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DEPARTMENT OF APPLIED SCIENCE

AN EXPERIMENTAL AND ANALYTICAL INVESTIGATION OF THE AERODYNAMICS OF A HYPERSONIC VEHICLE AT $M_\infty = 6$

by

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FINAL REPORT

NSG-1444

APRIL 1981

ABSTRACT

A summary of the work performed under NASA Contract NSG1444 is given within this report. Under this contract, the Antonio Ferri Laboratories at New York University performed experimental and analytical research on three-dimensional flow fields of a hypersonic cruising vehicle at a free stream Mach No. of 6. The goal of this research is the assessment of the validity of existing three-dimensional numerical programs in the prediction of the flow fields about general three-dimensional hypersonic bodies.

A detailed experimental research program was performed in which surface and flow field pressures were mapped. The results of the experimental work were compared with existing inviscid programs. Improvements have been made on the existing numerical methods to include angle of attack. The results of the above research have been reported in several technical reports, as well as in the open literature. A summary of this work is presented in this report.

INTRODUCTION

A continuing effort in updating the technology for high subsonic and supersonic aircrafts, Refs. 1-4, has concluded that the technology is available for an advanced supersonic cruise aircraft in $2 \leq M_\infty \leq 3$. These studies have also revealed the potential for significant advances in hypersonic technology that would back military and civil aeronautical capability needed in the future.

Over the past years, applications of hypersonic technology have been limited almost exclusively to the narrow field of space re-entry systems, Ref. 5. Computation of the space shuttle flow field, using noncentered finite difference schemes, has been developed, Ref. 6. A numerical procedure has also been developed in Ref. 7 to compute the inviscid hypersonic flow field about complex vehicle geometry. Although these programs have been generated for certain tasks, the applicability, testing, and shortcomings of these programs have not yet been identified. If a hypersonic vehicle is to be designed, it is important that we study the aerodynamic performance of such a vehicle, as well as the effects due to the perturbation of the configurations. In order for one to perform this task, and be able to identify regions where the program could give an improper result, it becomes necessary for us to rely on experimental verification. Such comparisons have not been performed extensively. Evidently, the prediction of this pressure field is a formidable task even from a purely inviscid point of view due to the three-dimensional effects introduced by both the body geometry, angle of attack, and roll. The situation becomes even more complex by the presence of important viscous effects which can have substantial influence on the

resulting pressure field.

In view of the above discussion, it is apparent that a computational procedure for predicting the aerodynamic characteristics of such hypersonic vehicles must include a very general inviscid flow field capability together with a procedure for making engineering corrections in the inviscid results to properly account for certain viscous effects.

With regard to current inviscid flow capability, it is noted that although several computational methods for cones at incidence exist, very few are capable of treating the very general situation which is of interest here. One of the schemes which shows promise is that due to Kutler (Ref. 6) which was developed by NASA Ames Research Center specifically for space shuttle application. This computer code has demonstrated the ability to provide accurate flow field description over complex afterbody geometries at moderate angles of attack. Shock capturing is assumed in this scheme by virtue of the fact that the working equations are written in full conservation form. The capabilities of this code at very large angles of attack ($\sim$ twice cone-half angle) have been examined by NYU personnel, and has been very successful in that:

a) no computational difficulties were encountered even at an angle of attack as large as 15° ($\alpha/\theta_c > 2$),
b) the presence of an embedded shock is well represented by the computed values of M_ϕ.

Although no computational difficulties were encountered, the accuracy of the scheme in terms of predicting actual pressure levels leaves much to be

desired. This can be attributed to the neglect of important viscous effects. The presence of these viscous effects has been recognized for a considerable time. However, it has been recently identified, (Ref. 9) and an effort has been expended to eliminate these deficiencies.

It is for the foregoing reasons that the New York University Antonio Ferri Laboratories performed research by testing a configuration of an advanced hypersonic vehicle in detail at a specific Mach number ($M_\infty = 6$) and a Reynolds number (4×10^7) in order to provide sufficient data for comparison with present available analytical programs.

Detailed measurements of surface pressure, static and total pressure profiles within the flow field external to an ogive-cylinder fuselage with swept back wing model were conducted with a Mach 6 blowdown type wind tunnel. With the aid of surface oil flow, the effect of angle of attack to the flow field, flow separation, vortex lift-off, and shock boundary layer interaction were identified.

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EXPERIMENT

The wind tunnel total pressure and total temperature were set at 600 psia and 980°R. A Mach 6.13 uniform flow was established within the test section. These tunnel testing conditions resulted in a free stream Reynolds number of 10^7/ft.

1. Model and Instrumentation

a) Model

A hypersonic cruise vehicle model, Fig. 1, was supplied by the NASA Langley Research Center. This model was mounted on a sting support and the model could be adjusted to have different angles of attack.

b) Instrumentation

Surface pressure taps were flush mounted over the model leeward surface and connected to two multi-port scanivalves. Each scanivalve was connected to 1 psia range pressure transducer. A pitot probe was used to measure the static and total pressure profiles within the flow field. The pitot probe was parallel to the model surface and transversed away from the surface in a direction perpendicular to the body. The static and total pressure outputs were connected to two separate pressure transducers. Outputs from the pressure transducers were recorded with a Honeywell 1616 visicorder.

SUMMARY OF RESULTS AND DISCUSSIONS

The research program was sub-divided in the following categorins:

a) Surface pressure distributions,

b) flow field measurements, these included pitot and static profiles,

c) oil flow studies,

d) inviscid analysis utilizing existing programs, and modifications if necessary,

e) identify regions of discrepancy between analysis and the experimental program, and

f) discussion of modified numerical program.

a) Surface Pressure

Leeward surface pressures over the wing fuselage afterbody have been measured with the model at $\alpha = 0°$, 5°, and 10°. A summary of the peripheral surface pressure distributions have been presented in Ref. 10, and a manual as well as tabulation of the surface pressure distribution has been presented in Ref. 11. In addition, the entire surface pressure measurements were placed on tape, and submitted with Ref. 11, on October 9, 1979.

b) Flowfield Measurements

Pitot measurements have been taken at angles of attack $\alpha = 0°$, 5°, and 10°. The stations and conditions are presented in Table I, which has been taken from Ref. 11. All the tabulations has been presented in Ref. 11, as well as

placed on a tape together with the surface pressure measurements, and have been submitted with Ref. 11, on October 9, 1979.

c) Oil Flow Studies

Oil flow pictures were taken on the model at $\alpha = 0°, 5°$, and $10°$ for both the Leeward and Windward side of the model. The results have been presented in Ref. 10, and submitted to NASA on March 29, 1979.

d) Inviscid Analysis

The three-dimensional supersonic flow field surrounding a space shuttle orbiter configuration has been analyzed numerically in Ref. 6. The governing hyperbolic partial differential equations were cast in conservation-law form and integrated from an initial data plane downstream using either a second-or-third order noncentered finite difference scheme. Similar numerical methods have been used to solve the inviscid flow field for re-entry vehicles with control surfaces in Ref. 7. This technique is used in the present study to analyze the inviscid flow field over the present configuration.

MacCormack's second-order numerical scheme, Ref. 8, was the first method of the four possible variations for replacing the space derivatives, was used. The grid size consisted of 14 radial and 39 meridional mesh points. Courrant number = 0.9 was chosen for stability control. Various clustering in the meridional place, with high grid point density near $\phi = 90°$, was used in order to obtain satisfactory accuracy. However, radial clustering dissipation and dispersion have not been considered in this research.

In the program, the model configuration has been approximated with polynominal curve fittings, as a function of the axial direction. The nose region Z < 1" was assumed to be a 14 degree hald angle conical body. From Z = 1 to 14", the geometry was approximated with an ogive body. The wing leading edge is approximated with a very small radius half circle. An inclined surface (slope = 1 degree) was used to compensate the boundary layer effect and the singularity along the leeward wing surface.

By incorporating these changes into the program, an analysis was performed for the particular experimental conditions. These results have been presented in detail in Ref. 12.

e) Discussion of Regions of Discrepancy Between The Analysis and The Experimental Program

Results of the present study shows that the numerical method is unable to compute completely the entire flow field at $\alpha > 0°$. The difficulties encountered are discussed in detail in Ref. 12, and are summarized below. Possible steps to improve the inviscid supersonic flow numerical method, in order to generate an inviscid flow numerical analysis cabable of predicting the flow field over a hypersonic cruise vehicle are also presented in Ref. 12.

The difficulties of the existing numerical method are:

1. Model Geometry Input
2. Convergence of the Numerical Method
3. Bow shock-wing interaction

Attempts have been made in order to provide guidelines to overcome some of the above difficulties. These include the equivalent body technique, which could provide solutions to the leeward wing-fuselage junction, the wing leading edge, and the windward wing-fuselage junction. All these methods have been outlined in detail in Ref. 12.

CONCLUSIONS

An experimental and analytical research program on a hypersonic cruising vehicle at a free stream Mach number of 6 has been performed. The results of the measurements have been tabulated and stored on tape for future use.

Existing numerical analysis of the inviscid supersonic flow theory was used to compute the flow field over the model. Comparison between the experimental results and the numerical analysis were made. Difficulties in utilizing the existing numerical method to predict the flow field are described. An inviscid numerical program is provided to calculate the flow field over general three-dimensional bodies at low angles of attack.

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TABLE I

PITOT PRESSURE RECORD NO.

$M = 6.25$, $P_\infty = 12$ mmHg, $Pt_\infty = 480$ psia, $\beta = 0°$

Z in*	X in	α = 0°	α = 5°	α = 10°
5.0	0.0	1		
12.5	0.0	2		
15.0	0.0		16,23*	
	1.5		17	
16.5	0	3,10*		
	1.6	4		
19.5	0.0			31,34*
	1.6			35*
	2.25			32
20.0	0		18,24*	
	1.5		19	
	1.6		25*	
	2.1		26*	
	2.5		20*	
	3.0		27*	
21	0.0	5,11*		
	1.6	6,12*		
	3.0	7,13*		
23.5	0.0			36*
	1.52			37*
	2.25			33
	2.75			38*

TABLE I CONT'D

Z in	X in	α = 0°	α = 5°	α = 10°
25.0	0		28*	
	1.6		21,29*	
	3.0		22,30*	
26.0	1.6	8,14*		
	3.0	9,15*		
28.5	0			39*
	1.52			40*
	2.75			41*

* THIS INDICATES WINDWARD SIDE DATA. OTHERS ARE ON THE LEEWARD SIDE

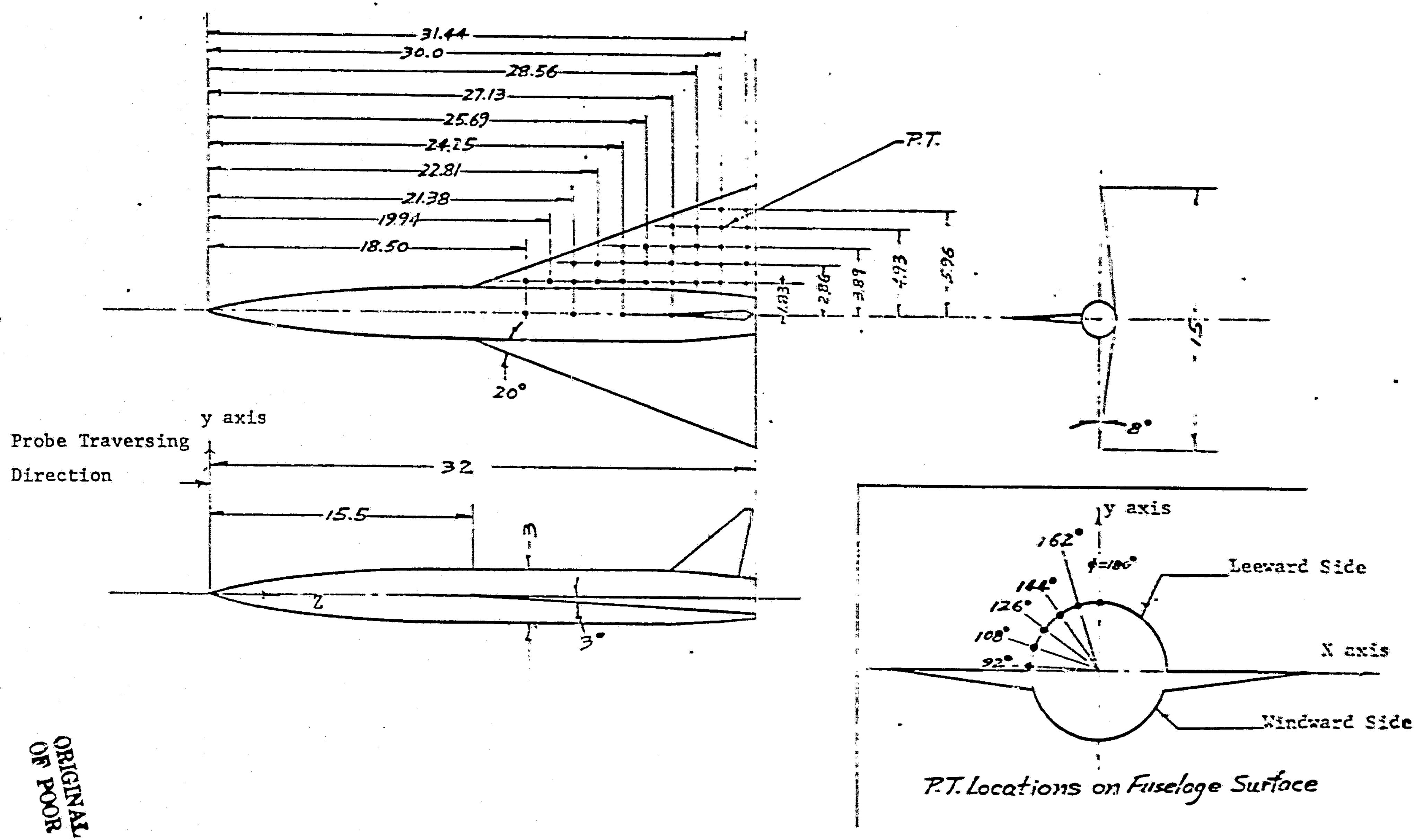

FIGURE 1

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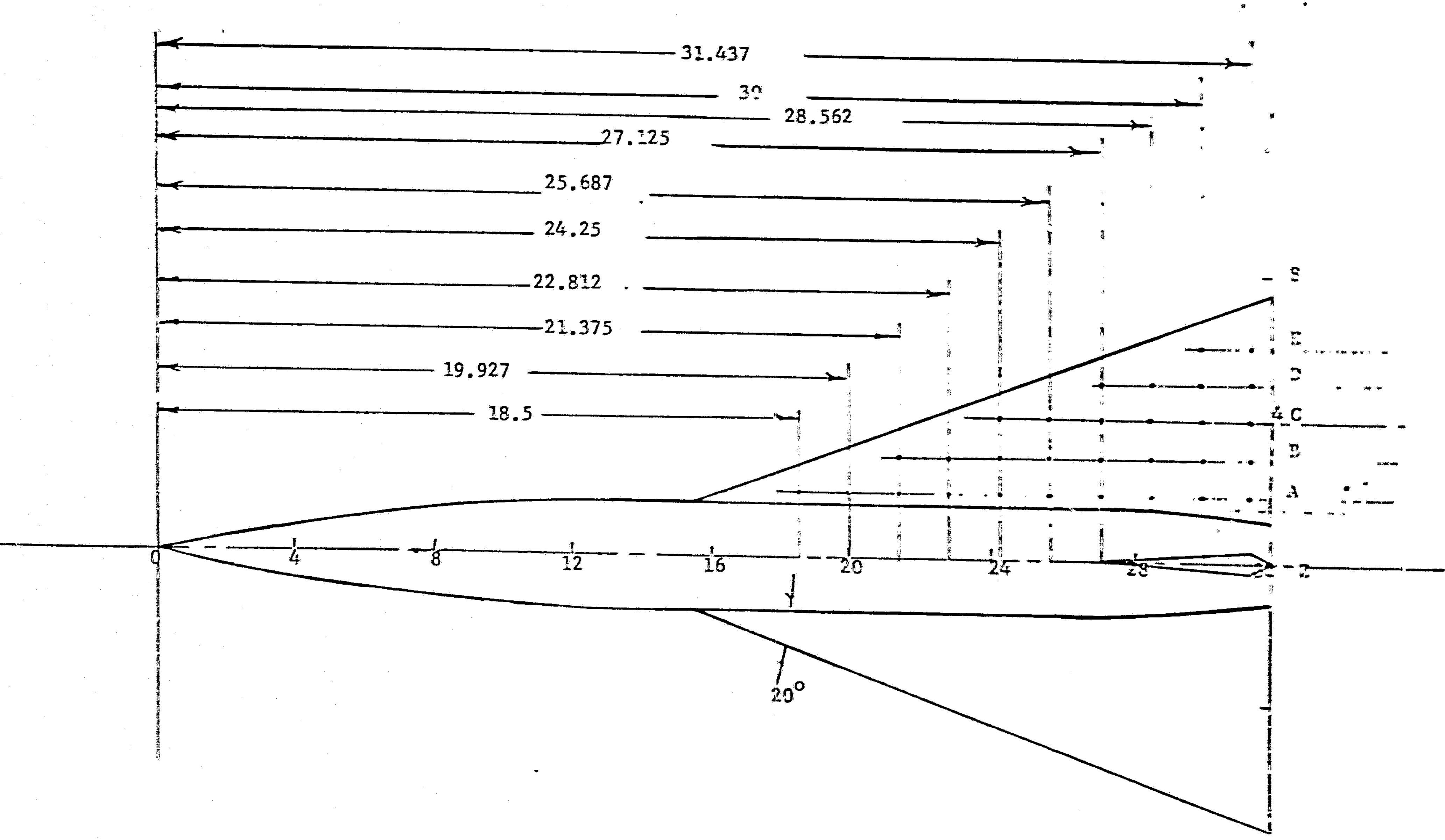

31.437
30
28.562
27.125
25.687
24.25
22.812
21.375
19.927
18.5
S
E
D
C
B
A
0
4
8
12
16
20
24
28
20°

FIGURE 1a

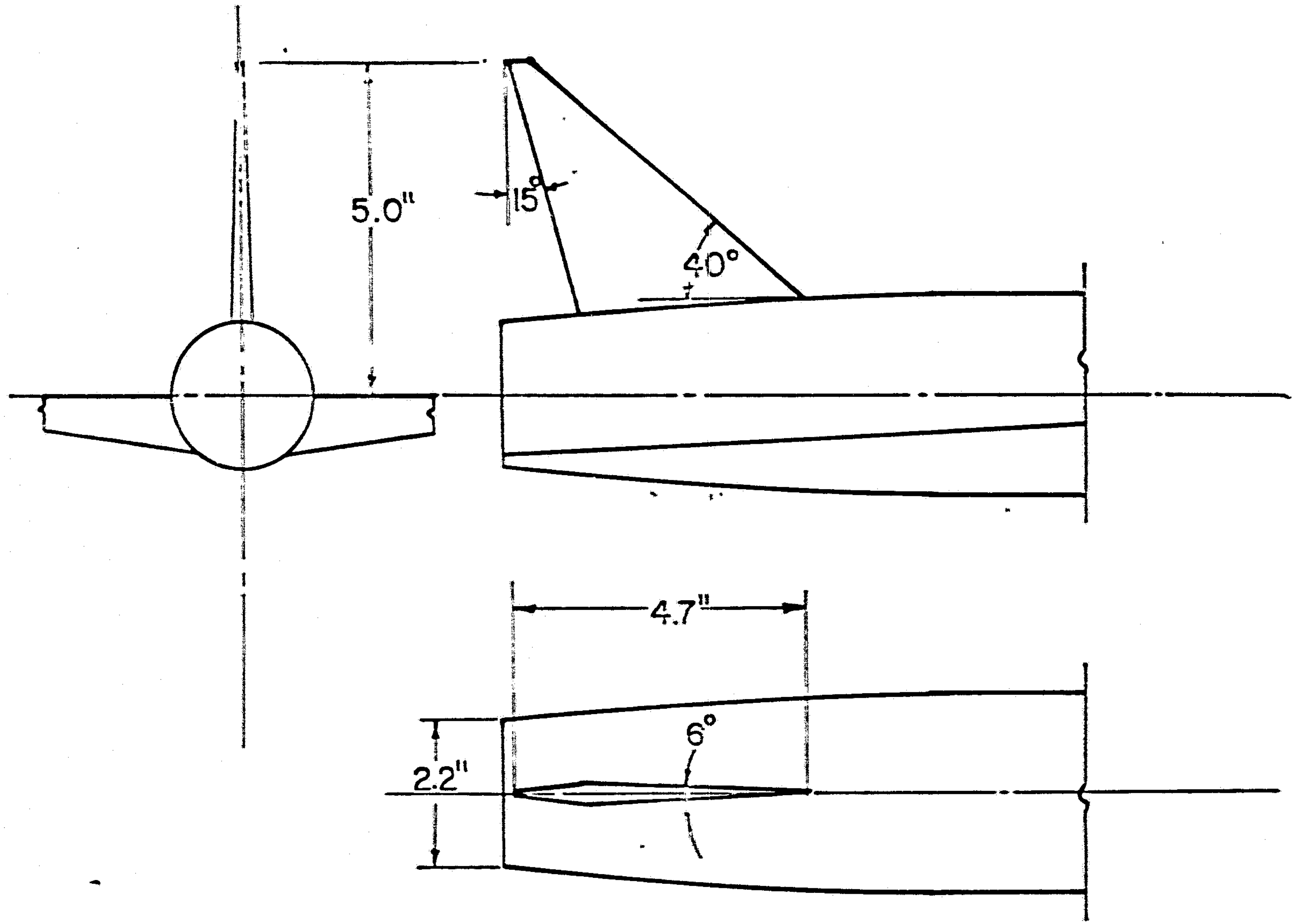
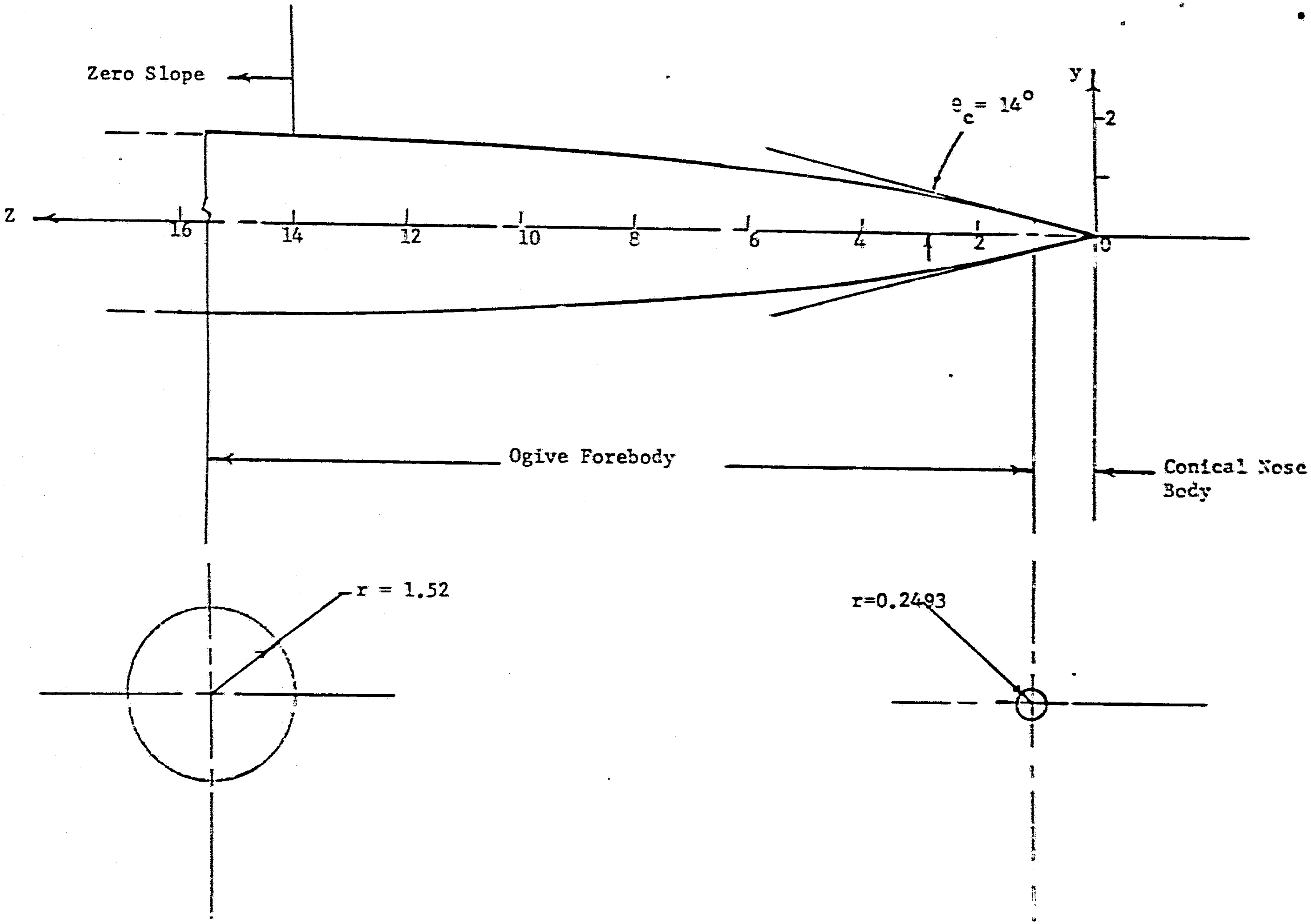


FIGURE 1b



Zero Slope
y
2
$\theta_c = 14^\circ$
Z
16
14
12
10
8
6
4
2
0
Ogive Forebody
Conical Nose Body
r = 1.52
r=0.2493

FIGURE 1c

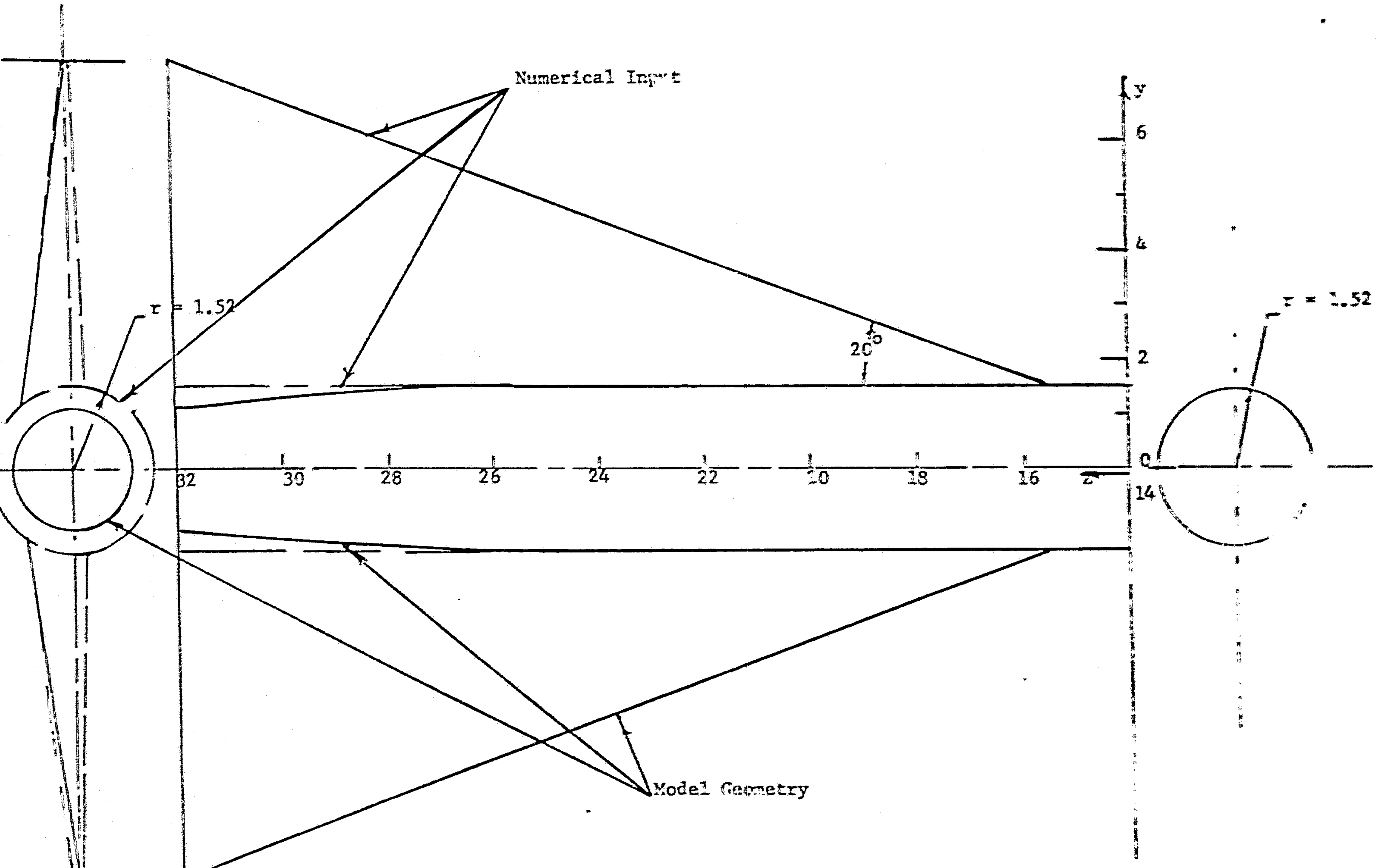
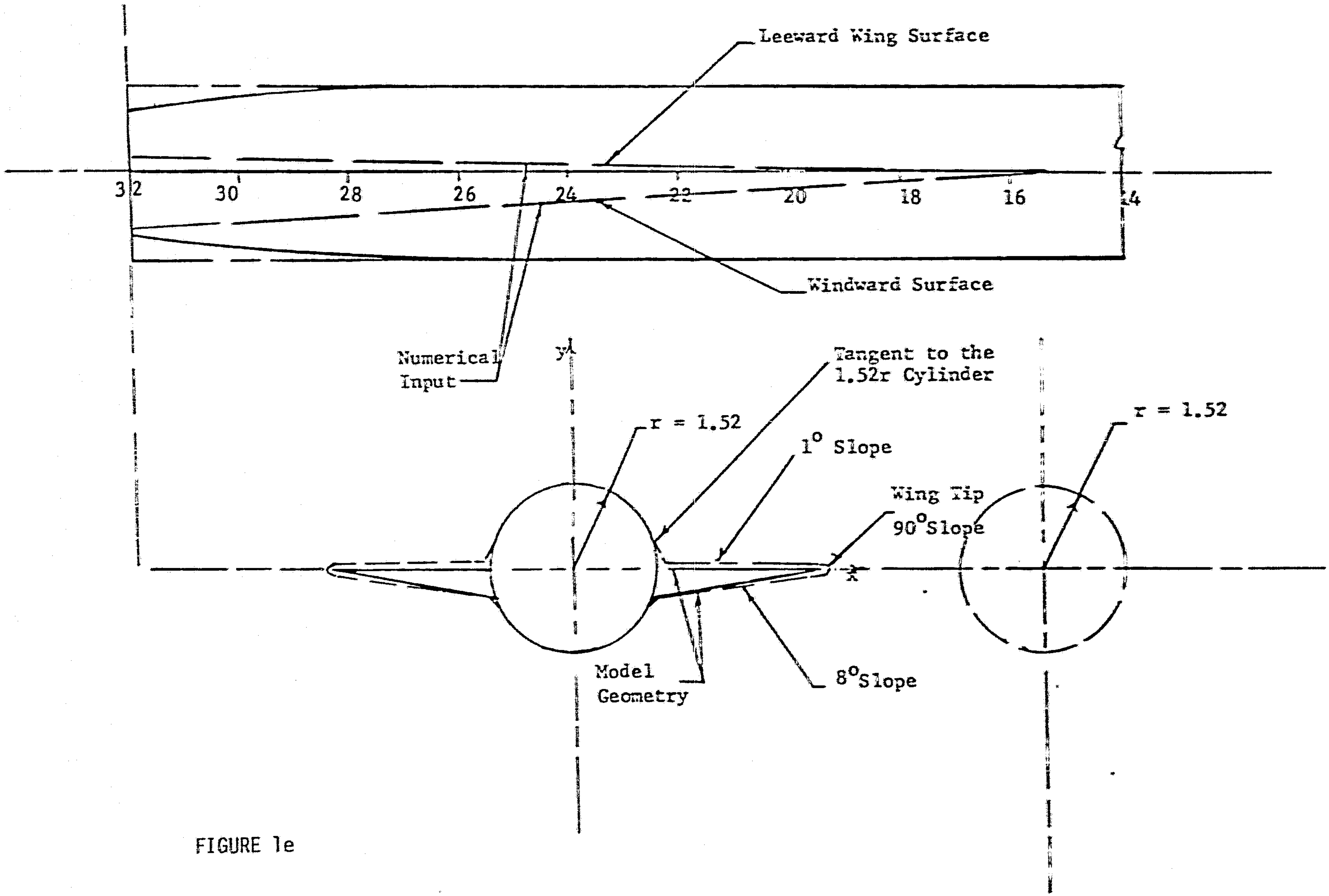

Numerical Input
Model Geometry
r = 1.52
r = 1.52
20°
y
6
4
2
0
z
14
16
18
20
22
24
26
28
30
32

FIGURE 1d



Leeward Wing Surface
32
30
28
26
24
22
20
18
16
14
Windward Surface
Numerical Input
y
Tangent to the 1.52r Cylinder
r = 1.52
r = 1.52
1° Slope
Wing Tip
90° Slope
x
Model Geometry
8° Slope

FIGURE 1e